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SPEED CONTROL OF A MISSILE WITH THROTTLEABLE DUCTED ROCKET PROPULSION

by

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SUMMARY

The demand for higher flight velocities and ranges for tactical missiles favours increasingly the application of airbreathing engines. The wide range of altitudes, in which many missiles - particularly air launched - must be operational, makes the variation of thrust mandatory and leads ultimately to the control of propellant mass flow. Thus, with the choice of a solid propellant ducted rocket as propulsion system, the gasgenerator must produce a variable mass flow rate. The widely applied method uses a propellant with a pressure sensitive burning rate. A valve with variable throat area controls the mass flow out of the gasgenerator. This report describes a mathematical model of the controller design.

Accordingly the mathematical model of the gasgenerator consists of the laws of propellant burning rate and chamber discharge and the fundamental equation of gas. This results in an opposing relation between mass flow and pressure for the first short control period after a valve throat variation. The controller design must consider this non-minimum-phase behaviour. For the presented example of a missile design a propellant mass flow measurement couldn't be realized. On the other hand the flight Mach number is the ultimate control value. This leads to the controller design described as follows:

The Mach number controller computes the suitable nominal value for the gasgenerator pressure while accounting for the pertinent gasgenerator dynamics. This procedure requires the knowledge of the pressure sensitive burning rate with adequate narrow tolerance limits. In case of high-g flight manoeuvres the system can optionally switch over to a simple command for the pressure nominal value.

An adaptive controller is nested within the Mach number loop to keep the gasgenerator pressure in closed loop control. Thus, an open loop control of the mass flow rate is provided. This loop is controlled by the flight Mach number loop, its value being derived from flight data.

The installation of prototype engines on test stands in a hardware-in-the-loop configuration and the required additional simulation procedure will be described. The presentation of test results concludes the paper.

LIST OF SYMBOLS

a	empirical parameter	Q	amount of heat
A	area	r	burn rate
c_p	specific heat at constant pressure	R	gas constant
c_v	specific heat at constant volume	S	reference area
c*	characteristic velocity	t	time
C_F	thrust coefficient	T	temperature
C_W	drag coefficient	T_r, T_p	time constants
F	thrust	v	velocity
g	gravity acceleration	V	volume
H	altitude	α	angle of incidence
m	mass	γ	ratio of specific heats
$\dot{m}$	mass flow	ρ	density
M	mach number	Θ	inclination angle
n	pressure exponent	ω_0	control parameter
p	pressure		

SUBSCRIPTS

b	burning
d	discharged
g	gas
p	produced
r	references
s	set value
t	throat
o	free stream

1. INTRODUCTION

The growing demand for higher flight velocities and ranges in the field of tactical missiles favours increasingly the application of airbreathing engines. The wide range of altitudes, in which many missiles - particularly air launched - must be operationel, together with requirements for improved manoeuvrability makes the variation of thrust mandatory. The most effective way to do this is the control of propellant mass flow.

With the choice of the propellant a strong preference of solid grains should be considered. A very attractive propulsion system which fits perfectly in this scenario is the solid propellant ramrocket or in the more usual american term the ducted rocket. The basic design of a missile with ducted rocket propulsion is shown in Fig. 1. The gasgenerator contains a grain of oxygen deficient propellant. After ignition it provides the ramcombustor with fuel rich combustion products. There they will mix and burn with the incoming air from the air inlets. The internal volume of the ramcombustor is suitable to accommodate an integral booster. For the most cases of air launched missiles the possible size of the booster is sufficient to accelerate to the required take over Mach number of the ducted rocket.

This paper describes the dynamic behaviour of the gasgenerator with the resulting design considerations for the control loop. A description of the complete engine control system shows the necessity to include some special features. The test philosophy with the according hardware-in-the-loop arrangement and the presentation of some test results will round off this paper.

2. GASGENERATORS WITH VARIABLE MASS FLOW

According to the requirements stated above the gasgenerator must produce a variable mass flow rate. This can be achieved in different ways based on the influence of the two main parameters

- burning rate
- burning surface.

A short survey of some proposed concepts shall be presented below (Fig. 2) without a discussion of the pros and cons.

o Pressure Sensitive Propellant
The widely applied method uses a propellant with a pressure sensitive burning rate. An end or cigarette burner is the common grain configuration. But also rod and tube grains are known. The control of the mass flow out of the gasgenerator and thus the gasgenerator pressure is managed by a valve with variable throat area.

o Matrix Propellant
This concept is a variation of the method before. The difference is given by the composition of the propellant. Preproduced granules of a propellant with low pressure sensitivity but high energy content could be embedded within a matrix of a low energy propellant with high pressure sensitivity. The result envisaged is a burning behaviour which resembles in some aspects a more or less erosive burning.

o Strand Augmented Propellant
The propellant composition of this method is somewhat reverse ot that above. One or more strands (small propellant "wicks") of high pressure sensitivity are embedded in a matrix propellant of high energy content but low pressure sensitivity. The now pressure dependent progress of the small burning front on top of the strands dictates how strong "coning" occurs in the matrix propellant. Coning means that a locally faster burn rate tends to deform at least the whole burning surface to a slope to this most progressive points. Therefore, the major effect is achieved by variation of the burning surface. The actuator nevertheless is still a throttling valve.

o Retractable Silver Wires
A burning surface modulation via controlled coning is the basis of this concept too. But differently from the method mentioned above the extend of coning is here influenced by the conductive heat flux through embedded silver wires. To maintain or stop the heat flux is the task of a somewhat complicated mechanism which retracts the silver wires at an adequate rate. This rate determines the relativ position of the silver wires to the burning surface and thus the corresponding heat flux.

o Longitudinal Tubes
A concept proposed by Thiokol as THERMATROL (R) (Thiokol Heat Exchange Rocket Motor Augmented Rate Control) uses the same effect of thermal induced coning but by other means. Instead of the retractable silver wires fixed small metal tubes would be installed. A flow rate control system connected to this tubes at the bottom of the grain allows a variable heat flux via the bleed rate of combustion gases.

As mentioned before a detailed discussion of the various concepts is not the aim of this survey. Nevertheless, one general aspect is very important to all control considerations. The variation of the burning surface is generally associated with relatively long time constants sometimes extending to a substantial part of the whole mission.

In the present case of an air-launched anti-ship missile the mission requirements postulated considerable versatility on the propulsion side. Thus, a gasgenerator with a pressure sensitive propellant became the heart of the speed control for the missile.

3. MATHEMATICAL MODELS

3.1 GASGENERATOR

A general arrangement of the selected type of gasgenerator is shown in Fig. 3. With the progress of the burning front to the left the free volume of the gasgenerator is filled with hot combustion products. This gas leaves the gasgenerator through the throttle-valve. The interaction of production and discharge of gas determines the pressure inside the free volume. A few words should be added referring to the nature of the combustion products. Beside their dominant gaseous portion these often contains liquid and solid phases. In particular with the application of propellants with high energy content the amount of these phases are not negligible. Nevertheless, with an assignment of suited material constants the mathematical treatment as an ideal gas leads to results of sufficient accuracy.

As mentioned before the conditions inside the free volume result from the difference between produced and discharged mass or energy respectively. Furthermore the volume of burned solid must be filled with gas. The burn rate of the propellant may be described by the known empirical formula

$$r = a \cdot p^n \tag{1}$$

The parameters a and n are propellant specific and not necessarily independent from pressure over the utilized pressure range as we will later see. The resulting mass production is given by

$$\dot{m}_p = \rho_p \cdot A_b \cdot a \cdot p^n \tag{2}$$

The mass flow discharged through the valve throat at a critical pressure ratio is, according to fundamental fluid dynamics, determined by

$$\dot{m}_d = \frac{p \cdot A_t}{c^*} \tag{3}$$

with

$$c^* = \frac{\sqrt{R \cdot T}}{\Gamma} \tag{4}$$

and

$$\Gamma = \sqrt{\gamma \left(\frac{2}{\gamma + 1}\right)^{\frac{\gamma + 1}{\gamma - 1}}} \tag{5}$$

The balance of mass is given by

$$\frac{dm}{dt} = \dot{m}_p - \dot{m}_d = \frac{d}{dt} (\rho_g \cdot V) \tag{6}$$

and corresponding the balance of energy

$$\frac{dQ}{dt} = \dot{m}_p \cdot c_p \cdot T_p - \dot{m}_d \cdot c_p \cdot T = \frac{d}{dt} (\rho_g \cdot V \cdot c_v \cdot T) \tag{7}$$

According to fundamental considerations /3/ temperature variations are small. Therefore, the assumption of constant temperature $T = T_p$ = const. is adequate.

With the basic assumption of an ideal gas the equation of state is valid

$$p = \rho_g \cdot R \cdot T \tag{8}$$

Thereby we obtain the differential equations of the gasgenerator

$$\frac{dp}{dt} = \frac{d\rho_g}{dt} R \cdot T = \frac{\dot{m}_p - \dot{m}_d - \rho_g \frac{dV}{dt}}{V} R \cdot T \tag{9}$$

$$= \frac{R \cdot T}{V} \left(\rho_p \cdot A_b \cdot a \cdot p^n - \frac{p}{R \cdot T} \cdot A_b \cdot a \cdot p^n - \frac{p \cdot A_t}{c^*}\right)$$

$$\frac{dV}{dt} = A_b \cdot r = A_b \cdot a \cdot p^n \tag{10}$$

With these equations the dynamic behaviour of the gasgenerator can be investigated. The parameter of interest is the discharged mass flow. Together with the air mass flow which is a function of the respective flight conditions this propellant mass flow determines the thrust of the missile.

An example illustrates this behaviour. As shown in Fig. 4 with a step function, the valve adjusting velocity assumed to be infinite, the throat area will be reduced. Now it is obvious that the mass flow shows a reverse-reaction behaviour. Due to the reduction of the throat area the mass flow is reduced proportionally for the first short period. This disturbance of the discharge causes the pressure to rise. But now, according to the rising pressure, the burning rate will also increase. Ultimately with a certain time constant presssure and mass flow will adjust to the new steady-state condition. The time constant is proportional to the free volume of the gasgenerator.

A very important aspect for the design of a controller is the question what kind of measurements are possible. With this the liquid and solid phases have to be considered. Of course, it is no problem to get a precise signal from a position indicator of the valve. But the corresponding throat area may depend on circumstances like burning time, pressure level etc. since the valve is exposed to deposits. With some precautions the measurement of pressures is feasible with sufficient accuracy. However, a reliable temperature recording of the hot gas is difficult and may depend on operating time.

Now, the possibility of a measurement of propellant mass flow is of dicisive significance. Extensive investigations /5/ have shown that out of a multitude of physical principles only a few are really applicable. For practical use in the environment which a missile provides there remains at least one method. This method makes use of a smoothing or intermediate volume served by the discharge of the throttling valve and an outflow through a fixed nozzle. Assumed there are no temperature variations and a critical pressure ratio is provided the mass flow is proportional to the pressure in the volume (Fig. 4). The deposit problem at a sonic nozzle should not be considered as a severe one. Nevertheless, a disadvantage of this method is a rise of the operational pressure level.

Besides the measurement aspects another point of strong influence is the behaviour of the controlling element represented by the throttle-valve with its actuator. Approximately with a first order lag element the transfer function of the throttle-valve can be described.

$$\frac{dA}{dt} = \frac{1}{T_v} (A_s - A) \qquad (11)$$

Furthermore must be observed

- the upper and lower limit of the valve throat area (according to design considerations it is impracticable to close the gasgenerator completely)

- the limit of the valve adjusting rate

3.2 MISSILE DYNAMICS

Related to the speed control the force balance along the flight trajectory is the significant portion of missile dynamics (Fig. 5). The resultant first order differential equation describes the acceleration along the flight trajectory:

$$\dot{v} = \frac{1}{m} \frac{\gamma}{2} p_0 \cdot M_0^2 \cdot S \cdot (C_F \cdot \cos\alpha - C_W) - g \cdot \sin\Theta \qquad (12)$$

The thrust coefficient is a function of:

$$C_F = f(\dot{m}_p, M_0, H, \alpha) \qquad (13)$$

and with a good approximation the drag coefficient is described by:

$$C_W = C_{W_0} + C_{W_\alpha} \cdot \alpha^2 \qquad (14)$$

where the basic term is determined by

$$C_{W_0} = f(M_0, H) \qquad (15)$$

Values of the drag coefficient are commonly provided by wind tunnel tests. Whereas with good accuracy the values of the thrust coefficient can be calculated by an engine cycle program.

The measurement of related flight data poses no problem. Practically each missile which such sophisticated requirements is equipped with an Inertial Navigation System (INS). Therefore, acceleration and velocity are available with great accuracy. Equally well it is possible to provide the Mach number by an aerodynamic probe.

4. DESIGN OF THE CONTROLLER

The considerations of the previous chapter points out that a design of direct or closed loop control of the propellant mass flow has to deal with the following aspects:

- reverse reaction or non-minimum-phase behaviour
- working pressure level of the gasgenerator is determined by two pressure losses (throttle-valve and sonic nozzle)
- intermediate volume has to be provided

On the other hand the parameter that ultimately must be controlled is the velocity or the Mach number. From this point of view it is obvious to reject the direct mass flow control and to attempt to design a control system which provides directly the required speed. With such a solution the propellant mass flow would be controlled in an open loop and the associated mass flow measurement can be rejected.

4.1 SPEED CONTROLLER

The requirements of the project mentioned in the beginning asked for

- Mach number control at different levels
- various flight profiles up to high altitude
- high-g manoeuvres at final approach to the target

To fulfill these requirements the following concept resulted from a somewhat difficult design process. The speed controller (Fig. 6) consists of a Mach number controller with a cascade controller for the pressure loop of the gasgenerator. Thus, an open loop control of the mass flow rate is provided.

At the high-g manoeuvres a control of the Mach number is no more opportune. In this case a set value adjuster will calculate the appropriate set value for the pressure controller from the required manoeuvre data. With that a sufficient thrust level will be provided to maintain a certain Mach number within specified tolerance limits during this flight period.

A short time before launch of the missile the required data must be transferred to the speed controller. Furthermore the interpretation of the signals given by a start sequencer and the timely start of the controller are properties of the initiation function block. Due to the transferred atmospherical and environmental data the applicable controller coefficients will be selected. The launch data decide the preadjustment of the throttle-valve. By this the appropriate mass flow rate for the transition phase will be provided.

In the next chapters both controllers will be described in more detail together with some design considerations. They are of the adaptive type. Thus, during the operation time a continuous parameter adaptation has to take place.

Furthermore the operation limits have to be taken into consideration. These limits consists of

- sufficient supercritical margin of the air inlets
  - minimum Mach number
  - maximum propellant mass flow
- heat resistance of radom
  - maximum Mach number
- strength of gasgenerator structure
  - maximum pressure inside the gasgenerator

A graph (Fig. 7) illustrates the remaining operational regime of the ducted rocket. During operation a supervisory function block determines the appropriate limitations.

Even without the discussion of the controllers in detail the above facts show that this task can only be done reasonably by direct digital control.

4.2 MACH NUMBER CONTROLLER

During the major part of the flight the speed control has to maintain a certain Mach number preset by the flight control system. A limiter controlled by the supervisory functions reduces the set value to $M_{smin} \leqslant M_s \leqslant M_{smax}$.

To improve the command behaviour this value will be treated with a filter. With the effected lag an overshoot of the Mach number progress will be virtually avoided.

A second limiter constrains the deviation to $DM_{min} \leqslant DM \leqslant DM_{max}$.

By this a violation of the specified Mach number limits due to the non-minimum-phase behaviour of the controlled system will be avoided.

At variations of altitude a fast adaptation of the set value of the gasgenerator pressure will be provided by a disturbance variable compensation.

The controller itself consists of a proportional-plus-integral controller for the processing of the Mach number deviation and a proportional controller. The latter stabilizes the loop with a feedback of d/dt(M) derived from the measured acceleration. The integral portion of the controller guarantees the static accuracy. Furthermore it provides a self learning capability. Therefore, the required gasgenerator pressure, which is a complex function of aerodynamic and environmental conditions, will be determined automatically.

The parameters of the Mach number controller will be adapted to the varying free volume and pressure inside the gasgenerator. The adaptation to the free volume, due to its slow variation, will be done continuously. However, the adaptation to the pressure has to consider that it is a state variable of the controlled system and will be done by parameter switching. At the switching points the continuity of the commanded value will be forced.

4.3 GASGENERATOR PRESSURE CONTROLLER

The considerations of chapter 3 demonstrated clearly that the pressure is the only variable of the gasgenerator which can be measured reasonably. It is described by the non-linear differential equation (9). From the viewpoint of the whole concept the purpose of the pressure control is primarily the stabilization of the gasgenerator. For it the behaviour of the pressure should correspond to the following reference model:

$$\frac{dp_r}{dt} = \frac{1}{T_p} (p_s - p_r) \qquad (16)$$

Thus, the transfer function of the gasgenerator will be linearized. Furthermore this function will be time-invariant. Consequently the design of the Mach number controller was simplified considerably. Due to the time-discrete realization of the controller and the limited valve adjusting rate the time constant T_p should be selected not too small.

The pressure set value can be provided by two modes:

- from the Mach number controller in the case of closed loop Mach number control

- from the set value adjuster in the case of open loop Mach number control at high-g manoeuvres.

The second mode is suitable for an open loop control of the mass flow (Fig. 9). At this the set value will be calculated with the known relation of mass flow and gasgenerator pressure (Equation (2) or tabulated data set). These conditions proved favorable at the design of the test concept.

The adaptive pressure controller (Fig. 10) consists of a proportional compensation controller with an additional proportional-plus-integral controller. According to Equation (16) the proportional compensation controller causes for the closed loop approximately the following transfer function:

$$\frac{dp}{dt} = \omega_0 \cdot (p_s - p) \qquad (17)$$

Without a violation of this behaviour the additional proportional-plus-integral controller provides the static accuracy. The portion of the correcting variable delivered by this additional controller can be limited. Thus, an excessive overflow of the integral value, e.g. caused by temporary jamming of the valve, with possibly resulting oscillations can be avoided. But with the increasing reliability of the valves this limiting functions lost its importance. Further will be limited:

- the set value of the pressure

- the set value of the valve throat area

both in accordance with the supervisory functions

- the valve throat area

due to design conditions.

Like these of the Mach number controller the parameters of the pressure controller will be adapted to the varying free volume and pressure inside the gasgenerator. The value of the free volume is known before ignition but there is no possibility to measure it during operation. Thus, it must be calculated continuously by the integration of the respectively produced increase of this volume. According to equation (10) the timely value is dependent on the gasgenerator pressure.

As indicated in chapter 3 the parameters a and n are not necessarily independent from the pressure. Typically the characteristic curve of the burn rate tends to minor gradients in the upper pressure regime. Therefore, the application of a tabulated form of this relationship and all parameters dependent of it proved favorable. The same is valid for the pressure dependence of the characteristic velocity c*.

During the development of the control system the test results suggested the inclusion of some special features. These accounts mainly for the behaviour of the particle-laden gas. Regardless of the measures taken at the valve the following provisions against the effect of deposits were made:

- identification of deviations from the valve characteristic curve
  At static operation phases continuous throat area deviations caused by deposits can be determined and compensated.

- wipe pulses
  At low pressure the preservation of the maximum throat area is particularly important. Wipe pulses at times can be helpful.

- switching of the loop gain
  At high pressure the control loop tends under some circumstances to oscillations. A reduction of the closed-loop gain in this regime solves the problem.

Furthermore a special initiation process provides for a smooth start and another feature opens the valve at burn-out for the largely utilization of the pressurized propellant gas.

5. SYSTEM TEST

Of course, an intense examination with simulations accompanied the design of the controller. At the same time the development of the engine takes place. For it the design of the throttle-valve was a very important matter. The various types with their pros and cons should not be discussed here.

Nevertheless, the applied and very approved design, a special kind of a rotary disk valve, shall be shortly presented. Fig. 11 shows a photograph of the valve seen from the gasgenerator side. The valve has four outlets in twos throttled by two rotary sliders with their axes perpendicular to the outlet plane. Shown is the full open position with the maximum throat area that can be provided. The control edges are shaped circular and rounded to form a nozzle. Thus, in the most critical position a geometry is provided which avoids largely deposition.

To obtain the above described state long series of tests were necessary. Of course, the early combination of the engine hardware with the control algorithms was very important. The resulting dynamic behaviour could be studied under real conditions.

5.1 HARDWARE-IN-THE-LOOP TEST TECHNIQUE

The test concept was strongly influenced by the following conditions:

- realization of the controller with direct digital control technique

- gasgenerator pressure control loop can be operated seperately

- necessity of propulsion system tests a long time before the availability of the controller hardware.

Under these circumstances a hardware-in-the-loop test technique offered a great advantage. Fig. 12 shows a comparison of the flight system with the conditions at the test plant. Due to the first point mentioned above the implementation of the controller software on a process computer instead of the engine control computer is possible without difficulty. At our test plant a VAX 750 computer was available for such purposes. Therefore, an operation of the (inner) gasgenerator loop at the test plant could be managed.

Although an electric drive is provided for the missile valve servo-system at the test plant a hydraulic actuator was used.

Much more complicated was the implementation of the Mach number controller. Therefore, a simulation of the missile behaviour due to inertia and aerodynamics must take place. Furthermore the test plant must be able to provide a varying air mass flow heated to the suitable temperature similar to the air inlets at supersonic flight.

Fig. 13 shows the situation in the form of a block diagram. The propulsion system is represented by

- the throttle-valve with the appropriate servo-system

- determining the mass flow out of the gasgenerator

- injected into the ramcombustor.

At flight conditions the achieved thrust acts against inertia and aerodynamics of the missile. The values of Mach number and acceleration measured by Mach meter and INS will be provided to the speed control system. The set value of the valve throat area closes the loop.

The algorithms of the speed control system can be performed either by the engine control computer or by a process computer.

As mentioned before the test plant must provide an appropriate air mass flow heated to the stagnation temperature. The available facility works according to the blow-down principle. The air mass flow out of an air reservoir will be controlled by a throttle-valve. A heater supplied with hydrogen provides the required temperature. Not shown on this diagram is the supply of the make up oxygen. This will be provided to an amount equal to that consumed by hydrogen.

However, the missing link for a successful test procedure is a substitude for the missile. This will be introduced by the simulation of the missile dynamics. Proceeding from the measured thrust all required parameters will be calculated. The corresponding simulation algorithms can also be performed by the mentioned process computer.

The block diagram of the simulation of the missile dynamics is shown in Fig. 14. Proceeding from the gas-generator pressure the produced mass flow will be calculated according to equation (2). With the integrated gas production the present missile mass can be calculated.

Simultaneously the coefficients of Thrust and Drag will be determined. The calculation of the fictive acceleration will be done according to equation (12). A first integration provides the velocity. Consequently the Mach number can be determined. Besides the preparation of the set values of the test plant controllers for air mass flow, heating hydrogen and make up oxygen the Mach number will be provided for the coefficient calculation of the next sample.

With an additional integration of the velocity times the cosine of the elevation angle the covered range over ground will be determined. The integrations will be performed with Runge-Kutta formulas.

Thus, the required values for the speed controller will also be provided.

5.2 TEST RESULTS

Fig. 15 shows a photograph of a ducted rocket propulsion system on the test facility. Lateral of the engine the hydraulic actuator of the throttle-valve is visible.

Results of two tests with the simulation of missile dynamics are presented in Fig. 16 and 17. Both show an acceleration process followed by a manoeuvre (open loop mode) in the second case. The diagrams on the left indicate the very good accordance of the gasgenerator pressure with its set value. The diagrams of the coefficients of drag and thrust show excess of thrust during the acceleration phase. The third set of curves illustrates the Mach number progress.

A photograph of a trial firing of a ducted rocket propulsion system (Fig. 18) closes this paper.

6. CONCLUSION

I think the now available combination of speed controlled airbreathing propulsion with a solid propellant offers very attractive possibilities not only for our current project but also for the whole spectrum of tactical missiles.

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Missile with Ducted Rocket Propulsion System
Basic Design

MBB Defence Systems Group

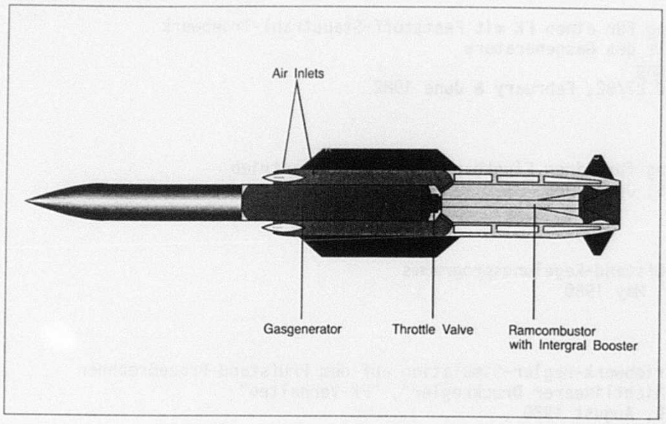

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Fig. 1

Gasgenerators
with Variable Mass-Flow

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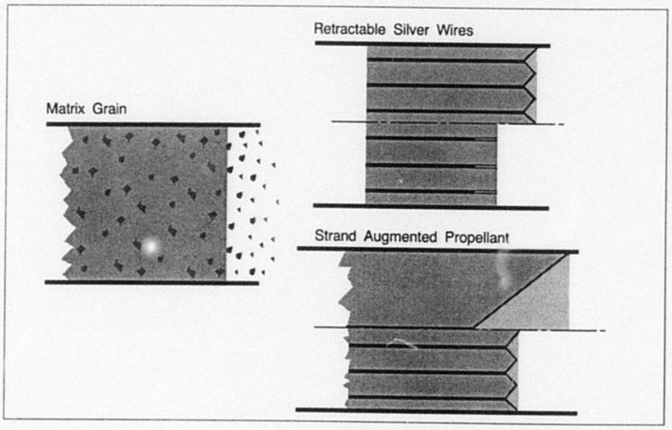

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Fig. 2

Gasgenerator
Pressure Determination

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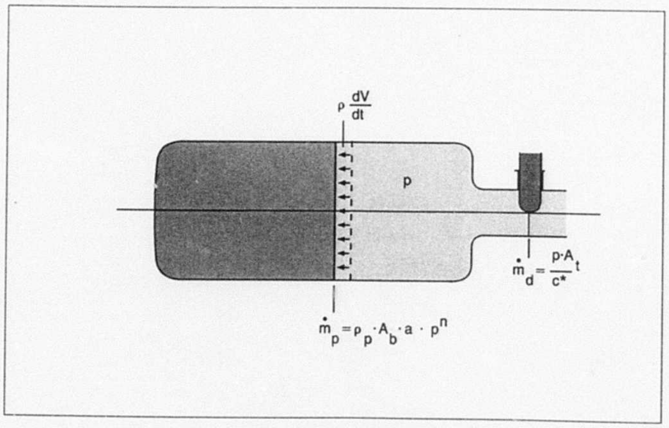

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Fig. 3

Gasgenerator
Non-Minimum-Phase Behaviour

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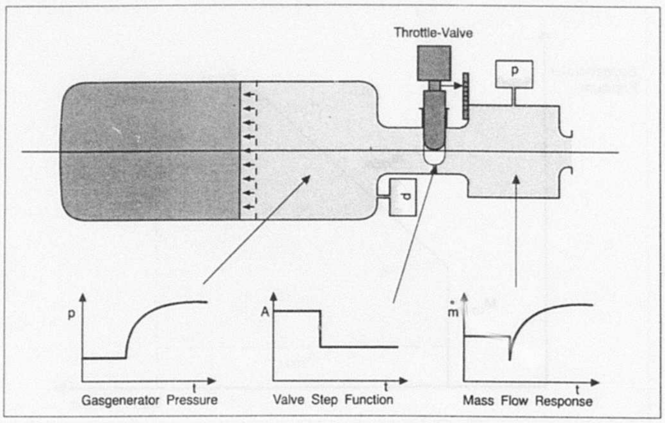

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Fig. 4

Missile Dynamics

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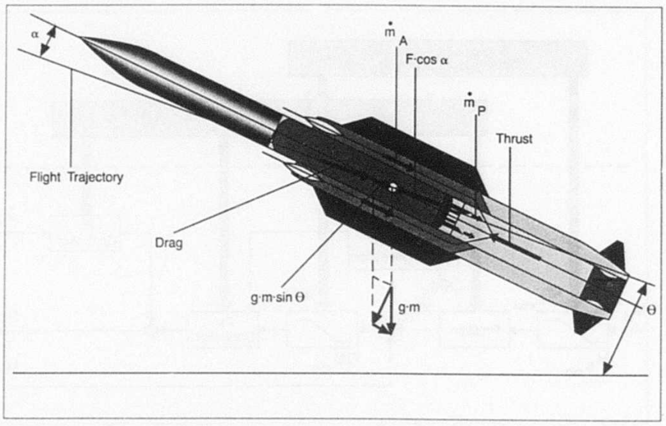

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Fig. 5

Speed Controller
Block Diagram

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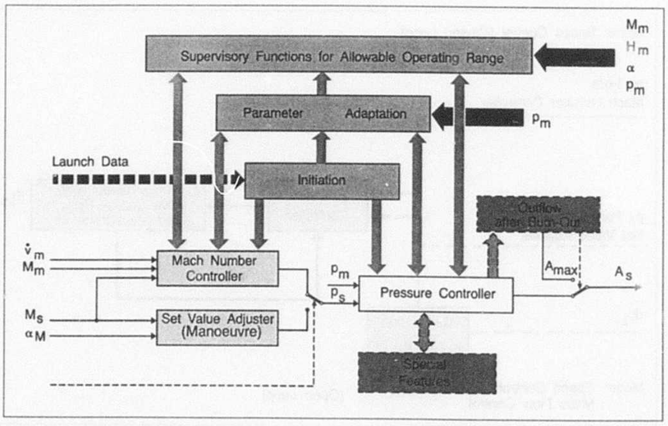

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Fig. 6

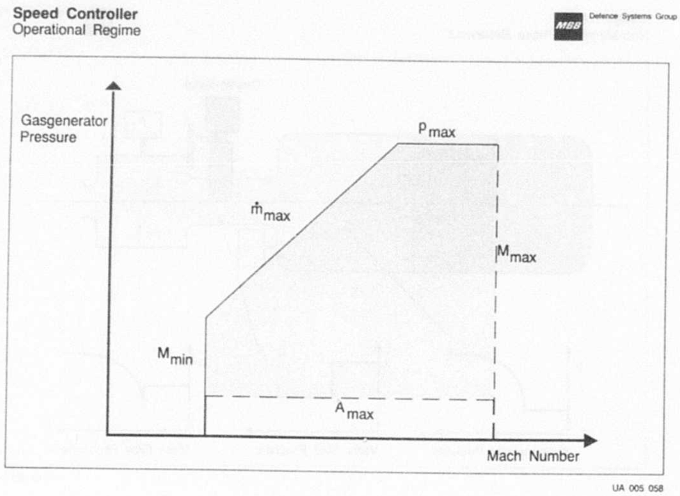

Fig. 7

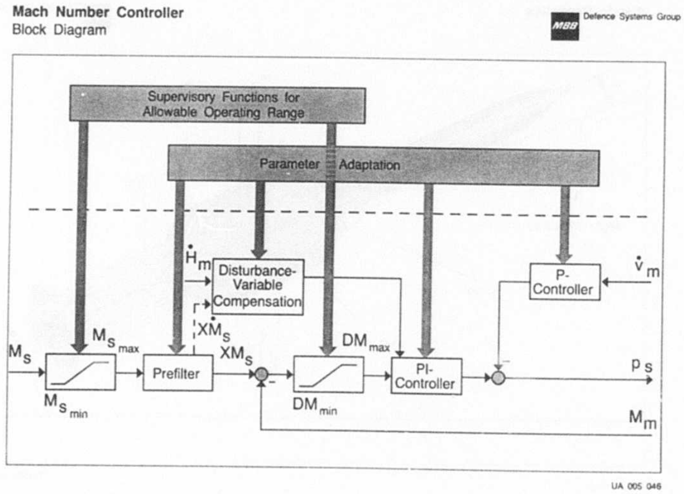

Fig. 8

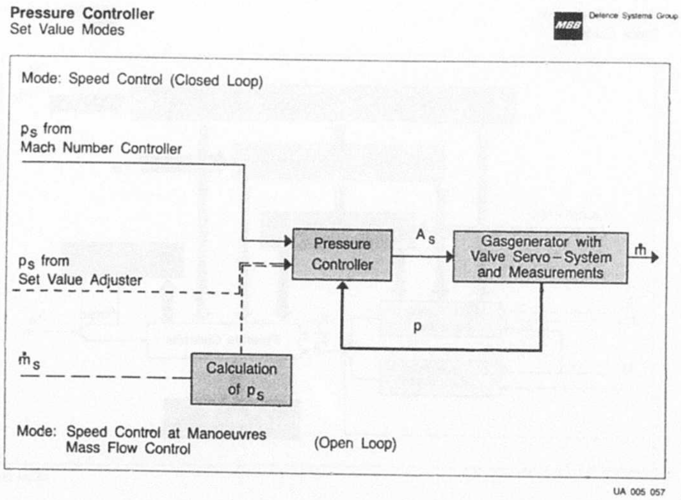

Fig. 9

Pressure Controller
Block Diagram

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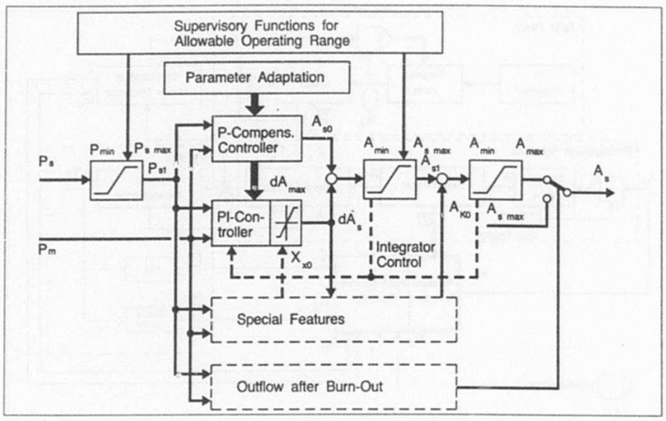

UA 005 054

Fig. 10

Throttle-Valve

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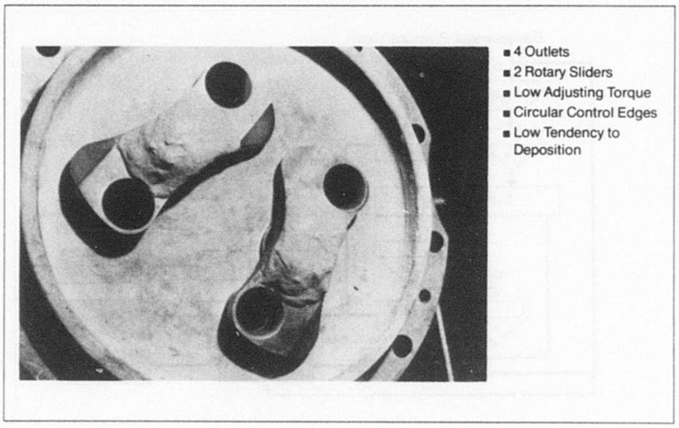
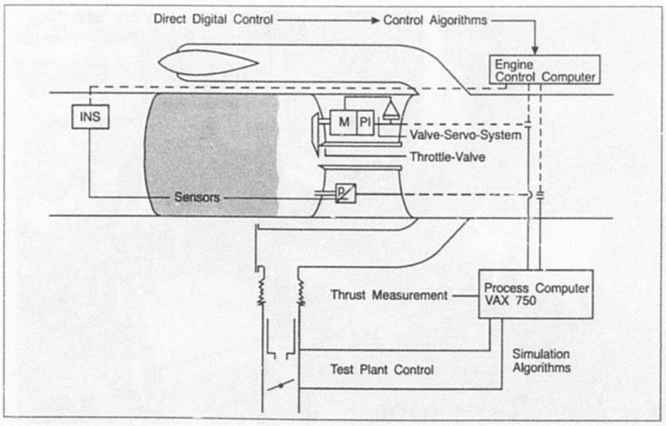


Fig. 11

Test Philosophy
Comparison: Flight System/Test System

MBB Defence Systems Group



UA 005 055

Fig. 12

Hardware-in-the-Loop Test Technique

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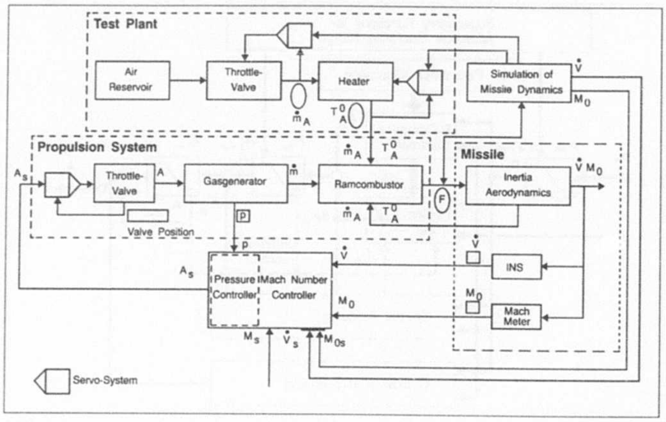

UA 005 056

Fig. 13

Simulation of Missile Dynamics
Block Diagram

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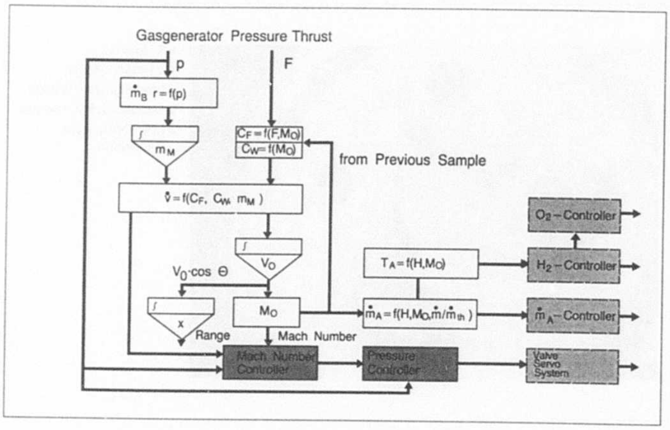

UA 005 048

Fig. 14

Test Plant with Ducted Rocket Propulsion System

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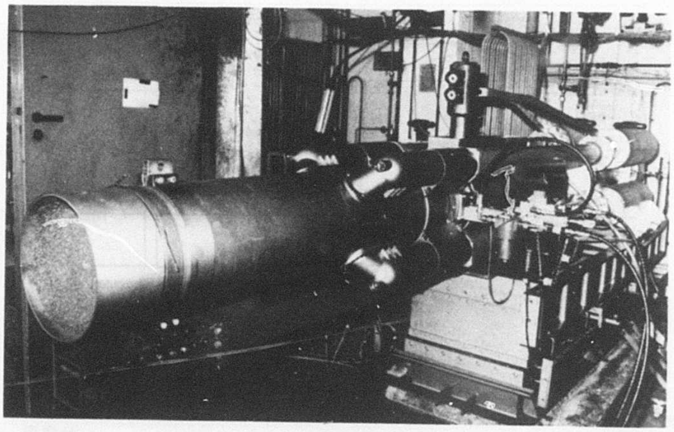

Fig. 15

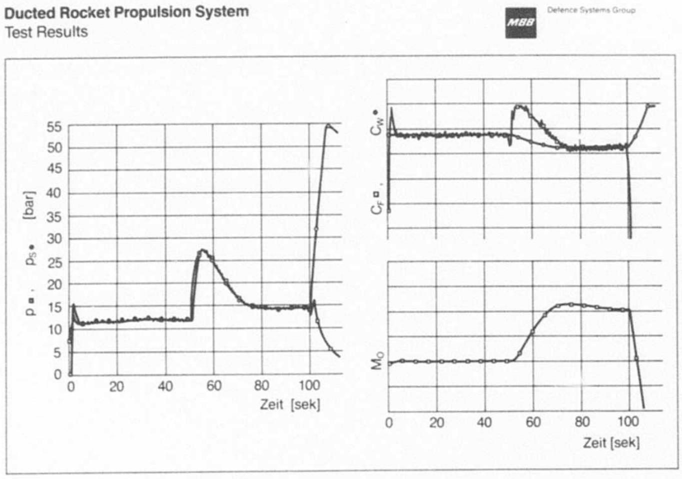

Fig. 16

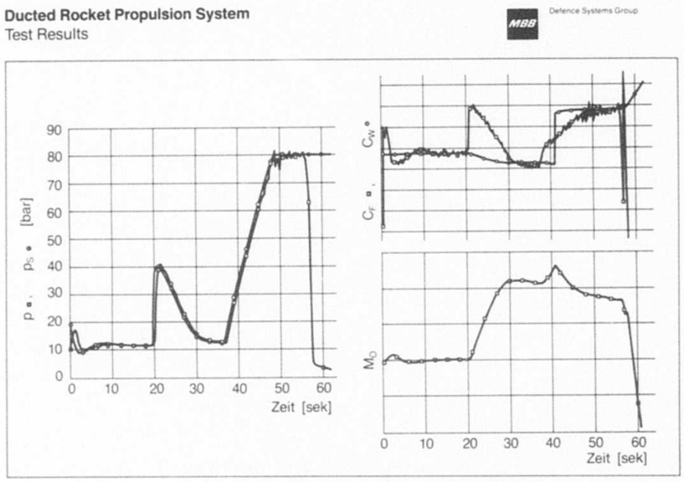

Fig. 17

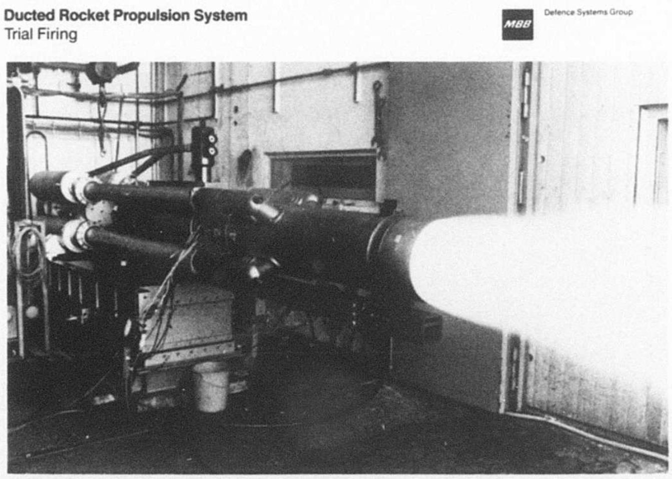

Fig. 18